AAS 95-111

NAVIGATION OF THE NEAR. EARTH ASTEROID RENDEZVOUS MISSION

J. K. Miller, W. E. 13011 man, R. P. Davis, C. E. Helfrich, D. J. Scheeres, S. P. Synnott, T. C. Wang, B. G. Williams and D. K. Yeomans †

The navigation of the Near Earth Asteroid Rendezvous (NEAR) mission is described. This mission is the first of several low cost missions being planned to study the solar system. The primary purpose of this mission is to orbit the near Earth asteroid 433 Eros and study it at close ra uge. Elements of the NEAR Navigation System are described including navigation instrumentation, the spacecraft attitude control and propulsion system and the ground system consisting of tracking stations and software. Navigation accuracies are given for spacecraft orbit prediction and control. Key navigation parameters are noted, including the physical parameters that describe Eros such as mass, moments of inertia and gravity harmonics. We also describe the orbit determination in support of science observations while in orbit about Eros.

INTRODUCTION

NASA's Near Earth Asteroid Rendezvous (NEAR) mission is planned to be the first Discovery-class mission and will be launched in February, 1996. The NEAR mission is managed by the Applied Physics Laboratory of Johns Hopkins University. The goal of this low-cost mission is to determine the physical and geological properties of a near-Earth asteroid and to infer its elemental and mineralogical composition by placing the NEAR spacecraft and its science instruments into orbit about the near-larth asteroid 433 Eros. The spacecraft will be launched from a Delta 11 rocket on a trajectory which takes approximately three years to rendezvous with asteroid 433 Eros in February 1999. Once in orbit about Eros, a sequence of maneuvers results in a 50 km circular orbit where the primary science observations will be performed. Twice during the ten month orbit phase, the orbit radius is reduced to 35 km for about 90 days total in order to obtain c.lose-in gamma and x-ray spectrometer measurements.

Navigation accuracies and propulsion system delta velocity requirements arc given in the paper for spacecraft orbit prediction and control. The orbit determination in support of science observations while in orbit about Eros is also described. 'l'he maneuver strategy and resulting accuracies are discussed for the interplanetary trajectory and approach to Eros. A AV budget is developed for the maneuvers required to accomplish these mission phases. Of particular interest is the sequence of maneuvers required to rendezvous and to establish the initial orbit about Eros. The data types used for orbit determination in support of these maneuvers include Doppler, range, and optical imaging of Eros. The strategy for initial acquisition of the asteroid with the spacecraft camera is described as well as orbit determination accuracies in support of the rendezvous burn.

The orbit phase of the NEAR mission presents many new challenges to Navigation. On previous missions[2–7] the optical data system errors have been dominated by the measurement error associated with the picture element resolution. Eros is expected to be highly irregular in shape and will be observed from several tens of kilometers with a data noise error of tens of meters. At this range, optical data system errors will be dominated by the ability of the optical data analyst to identify and locate landmarks. In order to determine the spacecraft orbit and certain physical parameters

† Members of the Navigation Systems Section, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California.

of Eros, the optics] data must be combined with Doppler data acquired from NASA's Deep Space Network (DSN). 'l'he combination of these two data types permits a three dimensional fix on the spacecraft which effectively removes singularities normally associated with Doppler data alone.

The effectiveness of Doppler and optical measurements in determining and predicting the spacecraft orbit is dependent on developing a precise physical model of Eros. The physical parameters that need to be determined include the mass, inertia tensor, gravity harmonics, and shape as well as the initial attitude and spin with respect to landmarks whose positions have been precisely determined with respect to the center of mass. Due to the expected irregular shape and weak gravity field, the gravity field determination presents a particularly challenging problem. The gravity field is modeled as a harmonic expansion of Legendre polynomials and associated functions. When the spacecraft orbit is several hundred kilometers from the asteroid, only the low degree and order harmonic coefficients may be determined. 'l'his suggests a strategy of lowering the spacecraft orbit to successively lower orbits and estimating higher degree and order coefficients as the spacecraft is maneuvered closer to the asteroid. Finally, at an orbit radius of 35 km, a sixteenth degree gravity field is required to adequately represent the asteroids mass distribution. Another challenging problem is the initial determination of the asteroid attitude, spin and inertia tensor. At the beginning of the orbit phase, very little *a priori* knowledge is available to describe the asteroid dynamics. A strategy is developed for determining this information.

NAVIGATION MISSION DESCRIPTION

The NEAR spacecraft will be launched by a Delta 11 rocket in February, 1996 on a 2-minus Delta-VEGA trajectory that will eventually lead to rendezvous with Eros in February, 1999. During the interplanetary flight after aphelion, the spacecraft will perform a critical deep space maneuver (DSM) designed to target the Earth flyby and then encounter Eros. The placement of the DSM within two astronomical units (AU) from the Sun was driven by the power margin available from the spacecraft solar arrays. The primary effect of the Earth flyby is to change the inclination of the spacecraft orbit to match that of Eros (10.8 degrees to the ecliptic). The Earth flyby also changes the spacecraft orbital energy to allow rendezvous with Eros. The interplanetary trajectory is illustrated in Figure 1.

Starting in January 1999, a sequence of approach maneuvers are planned to slow the spacecraft from an approach speed of about 950 m/s relative to Eros to a speed of about 5 m/s. The initial flyby of Eros will be on the sun-lit side at a altitude of about 500 km. Another sequence of maneuvers lowers the orbit radius to a 50 km circular orbit for the primary science observations. Since the orbit plane precesses relative to the Eros spin pole due to Eros oblateness and solar perturbations, there will be a series of orbit plane maintenance maneuvers occurring with a rnaximrrm frequency of about once per week during these orbits. However, twice during the orbit phase there will be favorable alignments of the Eros spin vector such that an equatorial orbit will persist without maneuver control. During these periods, the spacecraft will be maintained in an equatorial, circular orbit with a radius of 35 km for close observation of Eros, These two opportunities will allow a total of about 90 days in the low 35 km orbit.

During the orbit phase, there are mission design constraints imposed by the fixed mounting of solar arrays, science instruments, and high gain antenna on the spacecraft. These constraints impose limits on the orientation and subsequent motion of the orbit plane relative to the Earth and Sun direction. Maneuvers will be performed to maintain the orbit orientation within the spacecraft and science observation constraints. The type of maneuvers and their frequency will be a function of the orbit radius and of the Sun-Earth-Eros geometry over the ten month orbit phase. For the 50 km orbit, the angle between the orbit normal and Earth direction must be within 20 degrees, and for the 35 km orbit this angle must be within 30 degrees. In addition, the angle between the orbit normal and the Sun direction must be controlled to insure sufficient spacecraft power from the illumination of the solar arrays. For the first 100 days of the orbit phase, this angle will be less than 20 degrees. After this period, the angle will be controlled to be less than 30 degrees.

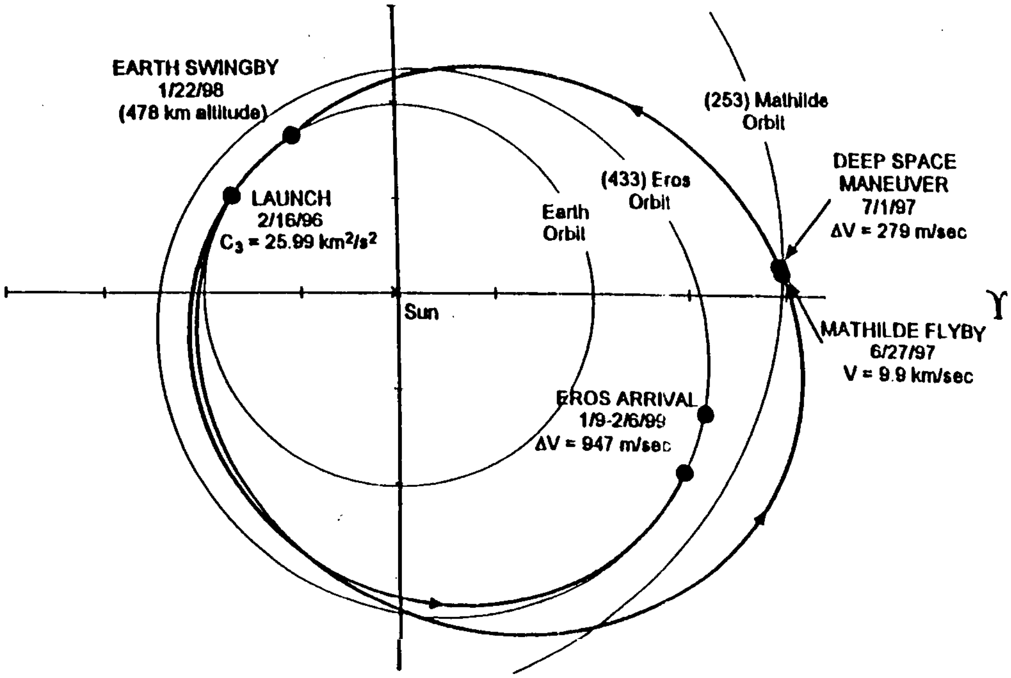


Figure 1 Interplanetary Trajectory

NEAR NAVIGATION SYSTEM

The NEAR Navigation System consists of a collection of hardware and software on board the spacecraft and on the ground whose function is to determine and control the flight path of the NEAR spacecraft. It consists of spacecraft instrumentation and DSN instrumentation that provides data for determining the orbit of the spacecraft, a propulsion system that is used to control the spacecraft's attitude and perform maneuvers, and software on the spacecraft and on the ground that is used to determine the spacecraft's orbit and to carry out it's navigation functions including command and control of the spacecraft trajectory.

Measurement **Subsystem**

The measurement subsystem consists of instruments that provide observations of the spacecraft's motion. instruments of this kind for the NEAR mission arc the spacecraft imaging system, coherently driven X-band transmitters and receivers, a laser altimeter, an inertial measurement unit containing three axis gyros and accelerometers and a star tracker.

Radio metric Data

Radiometric tracking data provide observations of the spacecraft motion with respect to the stations that comprise the DSN. The DSN tracking stations transmit radio-frequency signals to the spacecraft and receive signals via the spacecraft transponder and antenna. The received signals constitute observations of Doppler and range data. Doppler data provide a direct measure of line-of-sight velocity of a spacecraft relative to the tracking antenna. The accuracy of this measurement is about 1 mm/s at the S-band frequency and 0.1 mm/s at the X-band frequency when the two-way Doppler count is integrated for one minute. A single Doppler measurement provides no information on position and velocity normal to the line of sight; however, a series of Doppler measurements enables a precise determination of certain orbit parameters by observing the signatures due to the Earth's rotation and orbit dynamics in the data. The size, shape, and period of the orbit arc well determined and the orientation is marginally determined.

Range data provide a direct measure of the line-of-sight distance from an Earth tracking station to the spacecraft. The absolute range measurement is useful for determining the interplanetary spacecraft orbit and the ephemeris of Eros during the orbit phase, but is only marginally useful for determining the orbit of the spacecraft about Eros. For determination of the spacecraft orbit, the difference of successive range measurements is more directly useful. Differenced range may be obtained by differencing two range measurements taken over an interval of time or by integrating the Doppler data over the same interval. Since the integrated Doppler data are more accurate, this data type is used as the primary source of this information, However, range data provide the constant of integration that is needed for determination of orbits relative to the sun.

Optical imaging of Eros

optical imaging of Eros provides a powerful data type for aiding in the determination of the spacecraft orbit and describing certain characteristics of the asteroid. Optical data alone arc insufficient for complete orbit determination but are an essential complement to the Doppler data. When optical data are combined with Doppler data, an accurate determination of orbit orientation, size, and shape is obtained and the singularities normally associated with Doppler data alone are removed.

The optical measurement is obtained from an image of the asteroid using the Multi-Spectral Imager (MS]) which is a telescope with a charge-coupled-device (CCD) detector. The accuracy of this data type is a function of the picture element (pixel) spacing and the focal length of the camera optics. The MSI detector consists of a CCD with a usable 244 x 537 pixel sensor array. The camera field-of-view is 2.25° x 3° with a resolution of 95 μrad x 161 μrad per pixel. The camera focal length is 168 mm. This yields about 150 m resolution per pixel at the range of 1,000 km or about 7 m resolution in a 50 km orbit.

Onc characteristic of Eros that is important for orbit determination is the existence of landmarks on the surface[9–10]. In order to be useful for orbit determination, the landmarks must be readily identifiable by the human eye in images taken from different slant ranges and under a variety of lighting conditions. The existence of landmarks implies a surface that is varied in detail with some recognizable pattern, as would be caused by cratering or fracturing, for example. All of the bodies in the solar system with solid surfaces, whose surfaces have been observed, contain an abundant supply of surface features that may be used as landmarks for orbit determination. It is expected that Eros will also have an abundant supply. Previous flight experience has indicated that landmarks may be identified and located to an accuracy of about one pixel, for a pixel size of several hundred meters. In order to achieve this accuracy, the landmark must be identified as a unique surface feature by correlation with other surface features. This correlation may be performed by the human eye or perhaps by a computer program. At a scale of a few meters, it is assumed that the identification and location of landmarks may require several pixels of resolution. To be conservative, the pixel measurement error in the analysis that follows was adjusted to give a 20 m position measurement error at the surface of the asteroid.

Laser Altimetry

The NEAR Laser Rangefinder (NLR) provides a direct measurement of the range from the spacecraft to a point on the surface of Eros. The accuracy of this measurement is about 6m for an orbit radius of 50 km. When combined with a spacecraft orbit determined by radiometric tracking data arrd optical imaging of landmarks, the NLR data provides a high resolution determination of Eros topography relative to a set of control points that have been determined as a byproduct of navigation solutions for the orbit. The NLR data may then be combined with the precisely determined control points to obtain a precision shape model of Eros with a resolution of several meters.

In the event of an MSI failure, the NLR data may be used directly to determine the spacecraft orbit. The procedure would involve first determining a Doppler only solution for the spacecraft orbit and Eros attitude dynamics. An Eros shape model is determined as described above only without the optically determined control points to aid in the precision location of topography with respect to

Eros' principal axes. once a satisfactory shape model has been determined, NLR data arc processed in conjunction with radiometric data to determine the spacecraft orbit,.

Inertial Measurement Unit and Attitude Sensors

Orbit determination accuracy is directly dependent on the accuracy of spacecraft attitude determination and the accuracy of accelerometer measurements of spacecraft velocity changes during propulsive maneuvers and attitude thruster firings. Of particular interest is the determination of the direction in inertial space of the camera boresight at the time optical navigation images arc shuttered. For optical navigation, the direction of the camera boresight relative to the inertially fixed star field is determined on the spacecraft to an accuracy of 50 μrad and tagged to the shutter time to an accuracy of 20 ms.

A precise measurement of the velocity change associated with propulsive maneuvers is necessary in order to predict the trajectory of the spacecraft. A onc millimeter per second velocity error will map into a spacecraft position error of over a kilometer in 10 days. The Inertial Measurement Unit (IMU) has four accelerometrs that provide three axis measurements of acceleration. The fourth accelerometer provides redundancy. In order to provide the predictions of spacecraft positions following propulsive maneuvers that are needed for acquiring optical navigation images and controlling the orbit, the integrated accelerometer measurement must be good to a few millimeters per second.

Spacecraft **and** Maneuver Subsystem

The spacecraft contains elements that are considered part of the navigation system and assume troth active and passive roles. The propulsion and attitude control subsystems are active elements that provide the torques and thrust that controls the rotational and translational motion of the spacecraft. Other passive elements arc simply the source of nongravitational accelerations that tend to contaminate the navigation process and include attitude control gas leaks, thruster imbalance and the entire spacecraft that is accelerated by solar pressure.

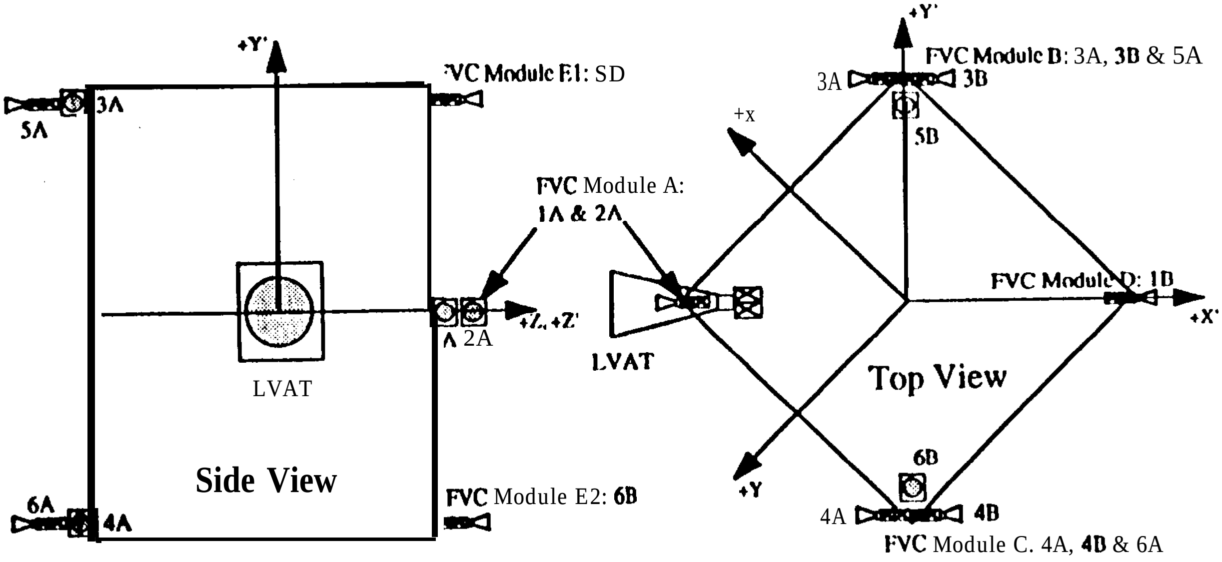
Maneuver Subsystem

The NEAR }'repulsion Subsystem uscs one N_2H_4/NTO bipropellant Large Velocity Adjustment (LVA) thruster, four N_2H_4 monopropellant Large Fine Velocity Control (LFVC) thrusters, and seven N_2H_4 monopropellant Small Fine Velocity Control (SFVC) thrusters. The LVA thruster delivers a 467 N force at 313 s specific impulse, the LFVC thrusters deliver 20.9 N at 234 s and the SFVC thrusters deliver 3.5 N at 226s. Propellant is contained in two oxidizer tanks and three fuel tanks. Spacecraft maneuvers arc performed using the thruster configuration depicted in Figure 2. These thrusters arc used for attitude control as well as propulsive maneuvers.

The current navigation specification calls for a 1σ total proportional pointing error of 6 milliradians (4 .24 per axis). The proportional magnitude specification is 0.2% (1 σ). These numbers apply to all interplanetary and rendezvous maneuvers. For the orbital phase, a fixed error of a few millimeters per second is needed to meet science requirements.

Spacecraft Nongravitational Accelerations

For an orbiting spacecraft, the orbit determination error is a function of errors associated with the measurement system and errors associated with modelling the dynamics of the spacecraft motion. Spacecraft dynamics are the direct result of forces acting on the spacecraft. These forces may be separated into two categories: gravitational forces arising from the central body and other bodies in the solar system and nongravitational forces arising from a variety of sources including solar radiation pressure and attitude control system gas leaks or thruster imbalance. The gravitational accelerations are determined by observing the motion of the spacecraft and, in the absence of a sufficiently sensitive accelerometer, the nongravitational accelerations arc determined the same way. The determination of the spacecraft orbit is thus dependent on the development of accurate models of the gravity field and the nongravitational force environment. The gravity field is generally easier to model in structural terms than nongravitational forces; however, the nongravitational forces are generally several orders of magnitude smaller. The stronger the gravitational forces and the weaker the nongravitational forces, the easier it is to determine the spacecraft's orbit.



1- **LVAT: 467 N @ 313 s**
4- **LFVC thrusters** (1A, 2A, 3A, 4A): 20.9 N @ 234 s
7- **SFVC thrusters: 1B, 3B,** 4B, 5A, **5B,** 6A, 69:3.5 N @ 226 S

Figure 2 NEAR. **Thruster Configuration**

Nongravitational accelerations that are constant may be modelled as bias parameters and are relatively easy to determine. Rapidly varying nongravitational accelerations tend to statistically average out over time. The most troublesome nongravitational accelerations are time varying at frequencies commensurate with the orbit period or tbc length of the data arc. The assumed *a priori* nongravitational accelerations associated with outgassing, attitude control gas leaks, and solar pressure arc given in 'l'able 1.

Table 1
NONGRAVITATIONAL ACCELERATION **MODEL**
PARAMETERS AND UNCERTAINTY

Parameters	Nominal Values	Error (1-sigm
Mass	800 kg	0
Attitude control gas leaks acceleration		
constant bias (km/s^2)	0	**5.0** x 1
variable modelling error (km/s^2)	0	5.0 x 1
correlation time (days)	5	0
Solar pressure		
effective area	10 m^2	0
momentum transfer coefficient	1.5	0.15

Ground Operations

The navigation ground operations system includes a large number of activities that arc performed on the ground in support of navigation. 'J'his system performs functions including calibration of DSN data, prediction of DSN antenna pointing and spacecraft frequency, orbit determination, maneuver determination and precision trajectory propagation. Supporting technologies such as ephemeris development and station location determination arc also included in this system.

Calibration

The instrumentation that provides the measurements for orbit determination and the spacecraft hardware required to control the spacecraft attitude and perform propulsive maneuvers must be calibrated in order to perform with the accuracy necessary to satisfy navigation requirements. Doppler and range tracking data arc calibrated to remove the effects of media on the velocity of signal propagation. The effective troposphere path length changes as the spacecraft elevation angle changes during a station pass. A seasonal model is used to make range corrections on the order of 15 meters over a station pass. Another transmission media effect is due to the charged particle content of the ionosphere and space plasma emanating from the sun. The ionospheric. effect is also elevation angle dependent and has a large diurnal dependence. Space plasma effects are less important, but may become quite significant at times of high solar activity or near conjunction at small Sun-Earth-Probe (SE]') angles. The DSN uses various sources of information to correct for the effects of media. These include tracking earth orbiting spacecraft including GPS satellites, photo polarimeters and water vapor radiometers.

Orbit Determination

In order to obtain an estimate of spacecraft and asteroid states, the calibrated radiometric tracking data and optical data are processed by navigation software using sequential filtering techniques[11-12], The estimates of the spacecraft state and asteroid state which includes attitude, arc then used to produce high precision trajectories and attitude predictions that are written to files and provided to other elements of the NEAR mission operations. in particular, these predictions arc used by the Mission Design Team to plan trajectory corrections and compute science instrument pointing and by the Science Team to plan science observations.

Maneuver Determination

Maneuvers are usually determined in a manner that minimizes total propellant consumption while satisfying certain mission constraints. This goal is accomplished through increased use of multi-maneuver trajectory optimization software, and proper Trajectory Correction Maneuver (TCM) planning (placement of TCMs, selection of aimpoints, etc), and through accurate design of each particular ΔV that is to be executed by the spacecraft[13]. Accurate delivery of the spacecraft to each target is an important goal for two primary reasons. In the case of a gravity-assist flyby, small delivery errors are amplified into large errors manifested at the next encounter. As a result, it is important to minimize delivery errors in order to achieve the goal of minimizing propellant usage. An accurate maneuver is also critical to the planned onboard science observation sequences. These science observations arc planned in advance and depend upon the actual trajectory nearly duplicating the flyby geometry assumed in the design process. Spacecraft navigation involves many statistical uncertainties that complicate the design of the perfect TCM. A TCM implements the ΔV that will correct an estimated state such that the spacecraft is placed on the desired trajectory for an upcoming encounter. The estimated state is obtained through the orbit determination process. The solution state has a statistical uncertainty, described in terms of a covariance matrix, based upon the consistency of the tracking data. This uncertainty is combined with the uncertainty in the location of the encounter body itself (ephemeris error) to produce the orbit determination uncertainty in the pre-TCM flyby conditions. This uncertainty describes the statistical errors expected from the orbit determination process. A second error source results from the uncertainties in the state of the spacecraft (tank pressures, temperature, thrust levels, etc.) at the time of the maneuver design.

Planet and Asteroid Ephemeris Development

Another activity within the Navigation System that supports orbit determination is ephemeris development. The J 1'1. ephemerides are produced by a system of programs that estimate the positions of the planets, asteroids, comets and satellites of the planets. Observational data are collected and processed in a differential least-squares program to improve the initial conditions of various bodies at some reference epoch. Starting from these improved initial conditions, the equations of motion are numerically integrated to produce ephemerides that are written to a file and provided to a variety of users including the orbit determination system.

The planetary ephemeris that will be used throughout the NEAR mission is the recently created JPL Development Ephemeris 400 (DE 400)[14]. DE400 utilizes the most recent values for the masses of the outer planets determined from the Voyager spacecraft flybys and unlike previous planetary ephemerides, DE400 is referred to the so-called IERS reference system.

DE400 will also be utilized in the ephemeris development of the rendezvous target 433 Eros[15-17]. Fortunately, Eros is one of the best observed near-Earth asteroids with 2477 optical astrometric observations available over the interval 1893-1993 and 4 additional radar Doppler and delay measurements available at the time of the Earth close approaches in January 1975 and December 1988. Prior to the NEAR spacecraft encounter, there will be additional ground-based observing opportunities in late 1995 and mid- 1998 to further refine the orbit of Eros. During the 1995 and 1998 observing opportunities, an effort will be made to reduce the astrometric observations with respect to the accurate star positions in the Hipparcos reference star catalog. Because of the long data interval and the powerful nature of optical and radar data taken near several previous Earth close approaches, the a *priori* ephemeris uncertainties are expected to be only about 50 km (1-sigma) at the time of the NEAR spacecraft encounter. Just prior to the encounter of the NEAR spacecraft, on-board spacecraft optical navigation images will reduce these uncertainties even further.

INTERPLANETARY NAVIGATION

The interplanetary phase is defined to begin at injection and end at the beginning of the approach to Eros. Table 2 gives a summary of the maneuver placement and rationale for the planned maneuvers of the interplanetary phase. There are two significant TCM's that are performed during interplanetary cruise. These are the first TCM performed seven days after launch and designed to remove injection errors associated with the Delta II rocket and a Deep Space Maneuver (DSM) performed near aphelion to shape the approach trajectory for the Earth flyby. TCM-1 has a mean velocity error of 9.8 m/s with a standard deviation of 10.0 m/s. The DSM maneuver has a deterministic velocity of 213.5 m/s and standard deviation of about 0.5 m/s. Other TCM's are small, but essential to achieve an accurate flyby of the Earth and an accurate rendezvous with Eros. The total mean ΔV for the other maneuvers will be on the order of 5 m/s.

Table 2
INTERPLANETARY **MANEUVER SCHEDULE**

Maneuver	Time	Date	Description
TCM-1	L+7d	Feb 24,1996	Needed to correct for injection errors
TCM-2	L+372d	Mar 3,1997	Deep Space Maneuver
TCM-3	DSM+30d	Apr 2,1997	DSM cleanup
TCM-4	E-60d	Nov 23,1997	First Earth approach
TCM-5	E-10d	Jan 12,1998	Second Earth approach
TCM-6	E+14d	Feb 5,1998	First Earth departure
TCM-7	E+120d	May 22,1998	Midcourse correction
TCM-8	Eros-60d	Dec 8,1998	Eros approach maneuver

Launch Phase

The launch phase begins at Earth injection and extends through TCM-1. The NEAR spacecraft will be launched by a Delta-II rocket. The first day of the launch window is February 17, 1996. The maneuver (TCM-1) at 7 days past injection is dominated by launch vehicle injection errors. Continuous Doppler tracking and a single range point from each station pass is obtained from the DSN for the first 10 days in support of TCM-1 and for recovery of the orbit. TCM-1 is optimized with the Deep Space Maneuver to reduce ΔV

Interplanetary Cruise Phase

During interplanetary cruise, the navigation operations become somewhat routine, The spacecraft is tracked by the DSN with two 4 hour station passes per week to maintain knowledge of the spacecraft trajectory. Repetition of passes from the same DSN station complex are avoided and coverage is spread around the globe to minimize the effect of station location errors. An occasional maneuver is performed to rc.store the spacecraft to the desired flight path.

Deep Space Maneuver

The current plan holds the DSM near aphelion, on March 3, 1997, and targets the spacecraft toward the Earth for a gravity assist en route to Eros. It is desired to keep the DSM AV direction at 90 degrees from the Earth-line to maintain telemetry during the maneuver. DSN tracking consists of onc 8 hour pass pcr day for three days before anti after the DSM and continuous coverage within 12 hours of the DSM. A clean-up of this maneuver, TCM-3, occurs 30 days after the DSM.

Earth Flyby

An Earth gravity assist is required to supply the necessary energy to reach Eros. The closest approach to Earth occurs on January 22, 1999, at an altitude of approximately 1200 km. Two statistical maneuvers occur prior to the flyby. The first occurs 60 days prior and the second occurs 10 days before the Earth closest approach. DSN tracking in support of the Earth flyby consists of onc 8 hour pass pcr day for onc month before and after the flyby and continuous coverage for seven days around the flyby. Canberra, Australia is the only visible station complex for 120 days after the Earth flyby because of the southernly direction of the Earth departure trajectory. Two statistical post-encounter maneuvers may be optimized to minimize AV while cleaning up Earth flyby errors.

EROS APPROACH NAVIGATION

The Eros approach phase begins at about 60 days prior to rendezvous with Eros. Navigation activities that arc performed during the approach phase include initial detection of Eros, search for co-orbitals, ephemeris refinement, and a sequence of approach maneuvers that are designed to reduce the spacecraft speed from about 950 m/s relative to Eros to a desired slow flyby of Eros at 500 krn on the sun side at about 5 m/s.

Initial Detection of Eros

The detection of Eros as early as possible is advantageous from the standpoint of ephemeris verification and improvement to assure early tracking in support of the initial approach TCM and rendezvous maneuvers, as well as for observing light curve characteristics and camera photometric properties (i.e. calibrations) with respect to the asteroid.

Detection depends, of course, on Eros's brightness as seen from the spacecraft and the imaging cameras sensitivity. The sensitivity of the camera depends on its light gathering capability (i.e. its aperture), the lens/filter/sensor light transfer and conversion efficiency, and the various electronic processes that produce image noise. The MSI is expected to have image noise levels less than one analog to digital quantization step level (i.e. quantization limited) when its CCD sensor is cooled to about -30 degrees C. The light transfer and conversion efficiency should be on the order of 20% for contemplated lens, filter and CCD quantum efficiencies. The MSI has a 63 rnm diameter aperture which will be fitted with a cover to shield the lens from propulsion ejecta. The cover reduces the aperture to 25 mm. With this cover on, initial detection can be expected at about 160 days from encounter (or 12.8 million km from Eros) when the asteroid will have an apparent magnitude of about 7.4 as seen from the spacecraft.

As the spacecraft/asteroid range closes, Eros's image will become brighter and expand thus improving the optical navigation image location accuracy. The asteroid's subtended angle will not equal the larger dimension of a camera pixel (161 microradians) until about 230,000 km relative range or well into the rendezvous sequence. However the image spread duc to refraction and charge dispersion within the CCD sensor will exceed one pixel well before this, resulting in improving image location accuracy as relative range decreases. At some relative range the accuracy of the optical data will begin to compete with the accuracy of the Eros's ephemeris uncertainty. For a conservative rss ephemeris uncertainty of 200 km and an optical image location accuracy of 0.3 pixels this cross over occurs at about 100,000 km relative range or about 3 days before the first rendezvous maneuver. A cluster of 8-10 pictures will be scheduled every week between 200 days and 17 days before closest approach for early detection of Eros and supporting maneuvers.

Approach Maneuvers and Orbit Insertion

The maneuvers which occur during the approach phase are given in Table 3. More than one maneuver is required because of the expected execution errors that will occur when performing a propulsive maneuver. For example, consider that the initial hyperbolic excess speed of the spacecraft is 1000 m/sec and the desired flyby speed is 5 m/see, but the expected spherical execution error is 1 %. If this slowdown is performed in one maneuver, then a 10 m/see spherical error can result, i.e. the spacecraft can be moving in any direction at any speed between O and 15 m/sec. If this hypothetical maneuver was performed near the asteroid, the spacecraft may move to a position relative to the asteroid that is unknown or may even result in impact.

Table 3
EROS APPROACH MANEUVER SCHEDULE

Maneuver	Time	Date	Description
TCM-9	Eros-28d	Jan 9,1999	First Eros rendezvous maneuver
TCM-10	Eros-21d	Jan 16,1999	Second Eros rendezvous maneuver
TCM-9	Eros-14d	Jan 23,1999	Third Eros rendezvous maneuver
TCM-9	Eros-7d	Jan 30,1999	Fourth Eros rendezvous maneuver

A series of three to four maneuvers is usually needed, each maneuver being 10% to 50% of the preceding maneuver, until the desired speed is reached. The number of maneuvers will increase as the size of the maneuver execution errors increase. In addition, the AV needed to compensate for these errors will also increase. Another factor to be considered is that a redetermination of the spacecraft's trajectory is needed after each maneuver, so that the next maneuver can be properly designed. A reasonable rule-of-thumb for time between maneuvers is 7 days. This allows for reconstruction of the previous maneuver, re-optimization of the remaining rendezvous sequence, re-estimation of the flyby altitude at the target body, design of the next maneuver, and the sequencing and up-loading process. Thus, in designing the rendezvous maneuver sequence, tradeoffs must be made between the total AV expended, the number of maneuvers, the required time between maneuvers, the total allowable time for the rendezvous phase, and the desired accuracy at closest approach.

For NEAR, the approach scenario includes four slow-dew]] maneuvers, each separated by 7 days. The first maneuver occurs on January 9, 1999, and brings the spacecraft velocity down from about 960 m/s to 250 m/s, The next maneuver uses what fuel remains in the bi-propellant tanks to reduce the velocity further to about 50 m/s. The third maneuver slows the spacecraft to 10 m/s, and the final approach maneuver, which occurs 7 days prior to the closest approach, brings the flyby velocity to 5 m/s. Closest approach is on February 6, 1999 at a distance of 500km.

EROS ORBIT PHASE NAVIGATION

Once the spacecraft achieves capture at Eros and the characterization and science phase of the mission begins, there are a number of mission design constraints which must be adhered to. These constraints drive the control of the orbit during this phase and place restrictions on what orbits are flown and when they are flown. A brief statement of these constraints follows: the spacecraft orbit should be safe and stable for a timespan of weeks, the spacecraft orbit normal should lie within a specified constraint angle of the Earth and Sun directions, normally there shall be no less than 7 days between maneuvers, the total mission AV expenditure shall remain less than 100 m/s, and the spacecraft shall orbit as low as possible (nominally at a 35 km radius) for as long as possible without violating any of the above constraints. The constraints on the spacecraft orbit normal are actually to be applied to the spacecraft orientation itself. However, assuming a nominally nadir pointing spacecraft, these constraints may be applied to the orbit normal.

These mission design constraints can be realized by controlling the spacecraft orbit inclination, node and radius[18-20]. The constraint that the spacecraft orbit be safe and stable during the mission duration is most easily met by specifying that the orbit always be retrograde with respect to the asteroid's rotation. Flying the orbit in this mode will usually ensure that the spacecraft will be in a non-synchronous motion state, and thus one needn't be concerned about the instabilities associated with direct orbits. To keep this constraint throughout the mission will require that the spacecraft orbit be changed by 180° around the mid-point of the mission (at a nominal cost of 8.3 m/s). 'I'his is necessary since the rotation pole of Eros lies near its orbital plane and sine.c the orbit normal must follow the Sun and Earth.

Forcing the orbit plane to comply with the two plane-of-sky constraints consumes the majority of effort during the orbital phase, and drives tbc mission profile. A plane-of-sky constraint angle of i_C forces the orbit normal to point within i_C degrees of the body (Earth or Sun) in question, thus defining a cone about the body vector. The problem is that tbe natural dynamics of the orbit plane about the asteroid will cause the orbit normal to precess about the asteroid's rotation pole, and usually will cause the orbit normal to leave the required constraint cone. Prior to this violation a plane change maneuver must be performed to reset the orbit normal within the constraint cone again. Ideally, the plane change maneuver will not change the orbit inclination, but will only rotate the argument of the ascending node, as measured in the asteroid equator,

The control of the spacecraft orbit radius is also a consideration. It n-rust be dealt with when transferring from higher to lower altitude orbits and when entering the near-circular 35km x 35km orbit. To support the design of these maneuvers, it is necessary to have a well characterized Eros gravity field and the proper targeting and modeling, tools to utilize these models, Note that when orbiting a body such as Eros, the use of osculating Kepler ian elements for targeting and orbit description is not well defined in general and use of such elements to design orbits and execute maneuvers could have negative consequences. ,

Combining the above mission design constraints, navigation needs and scientific concerns, a nominal mission plan has been developed which takes the mission from Eros rendezvous through December 31, 1999, the official end date of the mission. Table 4 summarizes the major events of the nominal mission plan and indicates briefly the rationale behind each phase.

Table 4
NOMINAL MISSION EVENTS TIMELINE

Description	Date	Day	Length	Orbit Radii
Eros C/A	2/6/99	O		
Characterize 2nd degree gravity	2/8/99	2	14 d	1000 km x 1000 km
Transfer	2/22/99	16	7 d	1000 krn + 200 km
Characterize 4th degree gravity	3/1/99	23	10 d	200 km x 200 km
Transfer	3/11/99	33	4 d	200 km → 50 km
Characterize 8th degree gravity	3/15/99	37	7 d	50 km x 50 km
Characterize gravity at north latitude	3/22/99	44	7 d	50 km x 35 km
Characterize gravity at south latitude	3/29/99	51	7 d	35 km x 50 km
Science phase mapping (low altitude)	4/5/99	58	52 d	35 km x 35 km
Science phase mapping	5/27/99	110	68 d	50 krn x 50 km
Characterize gravity at high latitudes	8/3/99	178	14 d	80 km x 35 km
Science phase mapping	8/17/99	192	6 d	55 km x 55 km
Plane flip maneuver	8/23/99	198		55 krn x 55 km
Science phase mapping	8/23/99	198	79 d	55 km x 55 km
Science phase mapping (low altitude)	11/'10/99	277	40 d	35 km x 35 km
Science phase mapping	12/20/99	317	11 d	50 km x 50 krn
End of Mission	12/31/99	328		

The total deterministic AV for this current, plan is 58,2 m/s. Note the periods of gravity mapping which occur during the mission. These periods arc essential to support the spacecraft's descent to a lower altitude. The most important of these periods occur in the weeks prior to the first

35x 35 km orbit period to support transfer of the spacecraft into that unprecedented orbit. The second period of characterization, when the spacecraft is nominally in a polar orbit, should enable the spacecraft to fly in higher inclination orbits during the second period of35x35km orbits. The timing of the 35x35 km orbits is driven by the geometry of the Earth, Sun and rotation pole of Eros. Should the actual rotation pole at Eros be significantly different from the current nominal value, the mission timeline may be significantly altered. The orbital altitude of the 50 x 50 and 55 x 55 km orbits are controlled largely by the 7 day minimum between maneuvers and the constraint angles to which the orbit normal must adhere. Should these constraint angles be relaxed, the altitude of these orbits could be dropped.

The navigation strategy required to determine the orbits described ahove and control the spacecraft is an integral part of the mission design. The data types include Doppler, range, and optical imaging of Eros. Of particular interest is the modelling of the optical imaging data type. For previous missions, the imaged objects have been large and nearly spherical and have been observed from relatively large distances. Eros is expected to be highly irregular in shape and will be observed from several tens of kilometers. At this distance it will be difficult to consistently locate a point in three dimensions that defines the center of figure when observations of only the lit limb and terminator, or portions thereof, are available. An alternative technique is to image a set of fixed landmarks on the surface of the asteroid and determine the orbit of the spacecraft directly by tracking these landmarks. Since the location of the landmarks in inertial space is dependent on the attitude of the asteroid, the orbit determination involves a joint solution for Eros attitude, landmark locations, and spacecraft position and velocity as well as other dynamic parameters including nongravitational accelerations and gravity model parameters.

A problem with the determination of the orbit of a spacecraft about an asteroid is the relatively large effect of nongravitational accelerations. These spacecraft accelerations are due to attitude control gas leaks, solar radiation pressure, and possible outgassing from the asteroid's surface. The latter effect acts on the spacecraft and Eros to produce both translational and rotational accelerations. Since the nongravitational forces and torques are time varying, a stochastic error model is required.

Physical Model of Eros

Determination of the spacecraft orbit about Eros is intimately associated with the development of an accurate physical model of Eros. Eros is the principal source of perturbations of the spacecraft's trajectory and the principal source of data for determining the orbit. The model of Eros used for orbit determination will be similar to the model used by the science team. The major difference is in emphasis of detail.

During a particularly close Earth approach (0.15 AU) in January 1975, there was a coordinated ground-based observation campaign to characterize the physical nature of asteroid 433 Eros[21–24]., Photometric, spectroscopic and radar measurements provided a diverse data set that allowed the asteroid's size, shape and spectral type to be well determined. The asteroid's shape can be approximated as a triaxial ellipsoid with dimensions 40.5 krn x 14.5 krn x 14.1 km and with a north rotation pole position (1950) given by an ecliptic longitude and latitude of 16° and 110 respectively. The dimensions are in error by 2 to 3 km and the pole is in error by a few degrees but there is no north-south ambiguity. Eros is an S type object with a geometric albedo of 0.16. From the light curve variations, which reach 1.47 magnitude in brightness, the rotation period has been determined as 5.27011 hours. The absolute magnitude of Eros (at zero phase angle and one AU from both the sun and Earth) is 11.16. Future ground-based observing opportunities in late 1995 and mid-1998 will be used to further refine the physical characteristics of asteroid 433 Eros.

The observed approximate shape has been embellished with craters arid surface features by the Applied Physics laboratory (APL) to define a reference model for navigation and mission design studies. This model is described by 4,202 vertices that are covered by 8,400 triangular plates. The parameters of the APL plate model are given in Table 5. For conservatism in the analysis that follows, the dimensions of this model are smaller than indicated above by most recent radar measurements.

For rotational stability, the polar axis or z-axis is perpendicular to the long axis and is the principal axis of inertia with the greatest moment of inertia. The long axis is therefore in the equatorial plane and is taken to be the z-axis and is the axis with minimum value for the moment of inertia. The y-axis completes the right hand body fixed coordinate system and is the axis with intermediate value for the moment of inertia or the unstable axis. Longitude (body-centered) is measured positive east from t he x-axis.

The mass properties and gravity harmonics given in Table 5 were obtained by numerical integration over the volume enclosed by the APL plate model assuming a constant density of 3.5 g/cm^3

Table 5
PHYSICAL MODEL OF EROS

Parameters	Values			
Size and Shape				
volume	3,790 km^3			
semi x-axis, y-axis, z-axis	16.7 km	8.6 km	6.3 krn	
Mass properties				
density	$3.5\ g/cm^3$			
mass	1.3×10^{16} kg			
GM	$8.86 \times 10^{-4}\ km^3/s^2$			
I_{xx}, I_{yy}, I_{zz}	22.9 km^2	63.9 km^2	70.9 km^2	
I_{xy}, I_{xz}, I_{yz}	0	0	0	
Gravity harmonics				
C_{20}, C_{22}	-3.0×10^{-2}	$+3.8 \times 10^{-2}$		
C_{40}, C_{42}, C_{44}	$+4.1 \times 10^{-3}$	-6.2×10^{-3}	$+5.1 \times 10^{-3}$	
C_{60}, C_{62}, C_{64}, C_{66}	-7.4×10^{-4}	$+9.9 \times 10^{-4}$	-8.7×10^{-4}	$+8.4 \times 10^{-4}$

Navigation Error Model

The study of orbit determination involves the development of models describing the spacecraft, the physical environment, and the instrumentation and data acquisition associate-d with 'orbit determination. These models are generally not worst case but represent a reasonably accurate reproduction of the system being investigated. Conservatism, if any, is introduced by assigning *a priori* errors that may be a little larger than their estimated values or by restricting the data set to a smaller number of observations than are available. orbit determination studies are performed that consist of computer simulations of a large number of cases varying the important system parameters. However, since only a limited amount of information has been gathered on the physical characteristics of Eros, the model that has been studied is probably considerably different from the actual asteroid. Therefore, the values of the parameters in this model have been biased slightly toward conservatism, but not far enough to be unrealistic..

The errors associated with the parameters of the orbit determination model may be separated into data noise errors, *a* priori' errors in the estimated parameters, and errors in the considered parameters. The data noise errors are the measurement errors and have been discussed above. The estimated parameters arc those parameters that are included in the orbit determination solution, and the *a priori* error is the initial error resulting from previous estimates of these parameters or other sources of information. In general, the *a* prior-i errors in the estimated parameters are set to very large values provided there is sufficient strength in the data to determine them. A notable exception is the nongravitational accelerations of the spacecraft. These include attitude control gas leaks, solar pressure and any drag that may be present associated with dust and outgassing from the asteroid surface. The nongravitational accelerations arc lumped together as a constant acceleration and as stochastic accelerations. 'l'he stochastic accelerations arc separated into two components and modelled as exponentially correlated process noise with correlation times of one day and five days, respectively. Other estimated parameters include the six dimensional state vectors defining the initial position and velocity of the spacecraft and Eros, another six dimensional vector defining Eros's attitude and spin, propulsive maneuver parameters, solar pressure parameters, and parameters that describe the physical characteristics of Eros. The Eros physical parameters include mass, principal moments of inertia, landmark locations, and gravity harmonic coefficients. A typical orbit determination solution may estimate as many as 500 parameters.

Some parameters cannot be modelled with sufficient precision to be estimated, and the systematic errors associated with these parameters are considered by the filter. The considered parameters arc DSN station locations and some of the high-order gravity harmonic coefficients. The sensitivity of the filter to considered parameters places a lower limit on the orbit determination accuracy. Of particular interest is the truncation error associated with the term.. of the gravity field expansion that arc not estimated. It can be shown that for a worst case mass distribution, a sixteenth degree and order expansion of Legendre polynomials and associated functions given by

$$U = kM\sum_{n=0}^{\infty}\left(\frac{r_o{}^n}{r^{n+1}}\right)\sum_{m=0}^{n} P_{nm}(\sin\phi)[C_{nm}\cos m\lambda + S_{nm}\sin m\lambda]$$

is adequate to model the gravitational acceleration to the measurement threshold for orbit radii greater than 35 km. The measurement threshold for Doppler data corresponds to a spacecraft acceleration of about 1.0 x 10^{-12} km/s^2. C_{nm} and S_{nm} arc the harmonic coefficients of the gravitational potential U, M is the mass of Eros, ϕ and λ arc the geocentric latitude and longitude respectively, and k is the universal gravitational constant.

Operational Orbit Determination Strategy

Immediately after rendezvous, very little is known about the dynamic and physical properties of the asteroid that would enable determination of the spacecraft's orbit with the precision required for navigation in the close orbits planned for the NEAR mission. Initially, the orbit determination strategy is therefore concerned with developing this information. The initial attitude and spin rate of the asteroid, as well as estimates of reference landmark locations, arc obtained from images of the asteroid. These initial estimates are used as *a priori* values for a more precise refinement of these parameters by the orbit determination software which combines optical measurements with Doppler tracking data to obtain solutions for the required parameters. As the spacecraft is maneuvered closer to the asteroid, estimates of spacecraft state, asteroid attitude, solar pressure, landmark locations and Eros physical parameters including mass, moments of inertia and gravity harmonics arc determined with increasing precision.

Eros Characterization

Determination of Eros' physical parameters, that are needed for navigation, will require several campaigns of intense observation each lasting about one week. During these campaigns, continuous Doppler tracking is required along with about 50 optical navigation frames per day. The first of these campaigns begins during the initial flyby of Eros and is concerned with determining the spin vector of Eros and identifying several hundred landmarks for navigation. The second campaign begins as the spacecraft enters an orbit radius of 200 km and involves combining the observed rotation of Eros with a solution for low degree and order gravity harmonics. The third campaign involves obtaining a solution for the free precession of Eros and is performed at the first opportunity depending on the magnitude of the spin vector offset from the principal axes. For a one degree free precession, this campaign would be initiated when the spacecraft is first maneuvered into a 50 krn orbit radius for detailed science observations. At this time a solution is also obtained for gravity harmonics through degree and order eight.

The first step in characterizing the physical parameters of Eros, is to take a few hundred images of the asteroid as the spacecraft slowly flies past Eros on the sun lit side. At closest approach of 500 km, the spacecraft cannot be turned to acquire optical navigation images because of Sun constraints. The images arc therefore acquired at a range of about 1000 km on approach to Eros and after encounter on departure. landmarks such as craters, boulders, and albedo markings arc identified as surface control points of Eros. A landmark database is then defined as a global control network of Eros. The control network of grids (landmark database) will be made denser as the image resolution increases. It will be used to process higher resolution images and to develop a shape mode] of Eros.

A rough solution for the spacecraft orbit is obtained from Doppler tracking data combined with images of Eros where the center of figure is located. This estimate is good to a few kilometers. The images of Eros obtained during approach arc inspected to find the image where Eros is extended to maximum length. In this attitude, Eros's axis of minimum moment of inertia is parallel to the focal plane of the camera. At the time of this image, a body fixed coordinate system is defined on Eros with the $x - y$ plane parallel to the focal plane of the camera and tbc z axis extended along tbc camera boresight. The x axis is placed along the long axis of Eros or principal axis of minimum moment of inertia. in this Cartesian coordinate system, the x and y Eros body fixed coordinates of reference landmarks may be determined directly from the camera photometrics and the range from Eros to the spacecraft which has been previously detcrrnincd by Doppler tracking. Only the z component of the reference landmarks and the location of Eros's center of mass in the reference frame defined above need be determined.

inspection of images taken a few minutes before and after the reference frame is defined (above) reveals an approximate direction and magnitude of the Eros spin vector. The estimated spin vector and landmark locations arc introduced to the orbit determination software and a solution is immediately obtained for these parameters relative, to Eros's center of gravity and in the coordinate frame defined above. If the free precession of Eros is less than one degree, a solution for Eros's spin vector and the spacecraft orbit may be obtained assuming principal axis rotation. Any free precession will be absorbed by the orbit determination filter and result in an apparent increase in the optical measurement noise. If tbc free precession is greater than a few degrees, it will be necessary to immediately solve for spin offset and principal axes moments of inertia. This procedure involves mapping the spin vector of Eros observed over several hours into Eros body fixed coordinates and observing the migration of the spin vector, From the observed dynamics and shape, enough *a priori"* information may be gleaned to obtain a solution for the parameters that define the free precession. In any event, a solution is obtained for the spin vector and the location of several reference landmarks.

The spacecraft is then maneuvered into a 200 krn circular orbit. At this orbit radius, the second degree gravity harmonics impress a distinct periodic signat ure in the Doppler tracking data. A combined Doppler and optical solution is obtained assuming principal axes rotation or if the free precession is large enough to have been previously determined, the moments of inertia and offset spin vector arc included. The optical data residuals are inspected and for tbc complete solution including free precession, the residuals will appear to be white noise with a onc or two pixel amplitude. If principal axis rotation had been assumed and the free precession is about one degree, the amplitude of the optical residuals will be about 10 pixels and modulated at a frequency dependent on the ratios of the moments of inertia, A solution may be attempted for the free precession and with some diligence it may be possible to isolate the spin offset from principal axis rotation and solve for the moments of inertia, If a solution is not obtained, the assumption of principal axis rotation will provide sufficiently accurate orbit determination to proceed to a lower orbit radius,

The spacecraft orbit will be eventually lowered to a 50 km circular orbit. At this orbit radius, the effect of onc degree of free precession will be greatly magnified and a solution for the free precession must be obtained since the assumption of principal axis rotation will not permit sufficiently accurate spacecraft orbit estimates. 'I'he previously detcrrnincd solutions assuming principal axis rotation will yield accurate landmark locations and the spin axis will serve as a close approximation to the direction of the angular momentum vector. From images of Eros taken at various directions, the semi-axes of Eros may be estimated and approximate moments of inertia computed from the equations for a triaxial ellipsoid. Also, the amplitude modulat ion of the optical residuals will provide some useful information to refine this approximation. An initial estimate of the spin vector is needed to complete the *a priori* information needed to attempt a solution. An approximate spin vector may be determined from several images spaced about 5 minutes apart. A complete solution for all the dynamic parameters of Eros may then be obtained including mass, moments of inertia and gravity harmonics. Solutions arc also obtained for the location of reference landmarks relative to Eros principal axes.

Orbit Maintenance

Except for a few weeks of intense navigation activity required to determine the physical parameters that characterize Eros, the navigation of the NEAR orbit phase is relatively benign. Data arc accumulated each day consisting of at least a pass of Doppler tracking data and several optical navigation frames. About every third day, a spacecraft orbit ephemeris and Eros attitude prediction arc written to a file that extends a minimum of 10 days into the future. 'J'bus, the mission design and science sequence teams will have an accurate estimate of the spacecraft orbit and Eros attitude for planning purposes. The spacecraft ephemeris and asteroid attitude files are also compressed and uploaded to the spacecraft for science instrument pointing control.

When a maneuver is needed to control the spacecraft orbit, the planned maneuver is included in the spacecraft ephemeris file. Maneuver execution errors of about 5 mm/s, which would occur if the spacecraft accelerometers arc unable to measure and control t he maneuver AV within 1 mm/s, will result in the actual spacecraft orbit deviating substantially from the predicted orbit. If not corrected, the desired objects of science and navigation observations, including landmarks, will migrate outside of the instruments field-of-view. Continuous Doppler tracking and several optical navigation images taken the day after orbit control maneuvers should be adequate to recover the orbit. A new spacecraft ephemeris is uploaded to the spacecraft within a day or two and science sequences may be executed as planned. Recovery of the spacecraft orbit is also aided by an optical navigation picture acquisition schedule specifically designed to allow a ncw higher accuracy landmark map to be made with every reduction in orbital radius of about a factor of two. The dense set of landmarks enables development of a large set of patterns that can bc used to uniquely identify locations on the surface. With the estimation, prediction, and display capabilty of navigation, the spacecraft location may be quickly determined.

in the event that the predicted spacecraft orbit migrates away from the nominal planned flight path by an amount that would jeopardize science observations or spat.ccraft safety, a snail vernier maneuver may be executed restoring the flight path to near the nominal.

Alternative Orbit Determination Strategies

A possible navigation system failure mode is loss of optical navigation data. If this failure occurrs early in the mission, it would bc necessary to navigate with radiometric data and any other information that may be gleaned from other spacecraft instruments and Earth based observations. Spacecraft instruments that arc useful for this purpose include the NLR and Near Infrared Spectrograph (NIS).

Determination of the spacecraft orbit with only radiomet ric data provides a baseline for evaluating enhancements that may be obtained by adding other data. Rendezvous would occur at 1000 to 2000 krn as before based on an *a priori* Eros ephemeris of 50 km from Earth based observations. With onc week of Doppler tracking, the spacecraft orbit relative to Eros should be known to 5 km and Eros gravity to about 1%. The spacecraft orbit is then reduced to 200 km as described above and the goal is to sense the spin and low-degree gravity harmonics. It may be necessary to explore other inclination orbits to sense principal axes if the *a priori*" pole is in error by more than five degrees or a large free precession exists. The spacecraft would dwell in this orbit for about 10 days and then be maneuvered into a 100 km circular orbit. In this orbit, the spin is refined and the gravity field is determined up to the eighth degree. The spacecraft orbit is incrementally lowered and circularized until the 35 krn circular orbit is achieved. 'I'his scenario will probably achieve the 35 km orbit at the second opportunity about 280 days after arrival.

The above "radiometric data only" scenario may be enhanced by using real time data from other sources. The NLR may be used to determine a shape model and then infer a gravity field based on the assumption of uniform density. This will aid faster gravity field recovery and spin determination. Later in the mission, NLR data may be used directly for orbit determination once a high resolution shape model has been determined

Another potentially useful source of data is the NIS instrument. A rough ellipsoidal shape could be determined during approach and aid initial acquisition. This rough shape would also be useful for aiding gravity field recovery and spin determination. Large co-orbitals may also be detected.

Finally, failure of the MS] would result in a more intensive Earth based observation campaign. An improved radar re-determination of Eros spin may be attempted during the Feb-Nov 1998 viewing opportunity.

Spacecraft Orbit Prediction

A detailed covariance analysis was performed to determine the predicted spacecraft orbit determination error at various epochs with respect to the end of the data arc or the time at which the last data point is acquired. Computer simulations of data scheduling, trajectory propagation, data filtering, and solution mapping were generated. The data arc spanned 10 days and included continuous Doppler and range radio metric tracking data and optical images of the asteroid. A propulsive maneuver was included one day before the end of the data arc with an execution error of 5 mm/s. Radio metric data were taken from Deep Space Stations 14, 43, and 63, located at Goldstone, California; Canberra, Australia; and Madrid, Spain, respectively. The Doppler data were compressed to one point every five minutes, and a single range point was taken from each station pass. The optical data rate was one frame every three hours. Estimated parameters included spacecraft state, Eros attitude, stochastic nongravitational accelerations, maneuver velocity components, principal axes moments of inertia, the locations of twelve landmarks and an eighth degree and order gravity field. Station locations and gravity harmonic coefficients above degree four were considered by the orbit determination filter. A simulated data set was processed by a square root information filter to obtain spacecraft state and Eros attitude prediction errors.

The spacecraft orbit determination errors are described in an orthogonal rotating frame with the x_s-axis directed radially from the center of the asteroid to the spacecraft (the *radial* direction), the y_s-axis normal to the radial direction and in the plane-of-mot ion (the *downtrack* direction), and the z_s-axis normal to the plane of motion (the *crosstrack* direction). For a circular orbit, the downtrack direction is along the velocity vector. Orbit determination results arc shown in Table 6 for circular orbits with periapsis radii of 200 km, 50km and 35 km. The radial, downtrack, and crosstrack orbit determination errors arc given for orbit prediction times of 0, 1, 3, 7 and 10 days from the time of the last data point. The orbit determination errors grow monotonically in semi-major axis of the error ellipsoid as the prediction time increases. Some individual components may decrease with time due to variations in the mapped true anomaly. The orbit determination errors also tend to increase as the radius of the orbit is increased, presumably due to the decreasing strength of the Doppler data.

Spacecraft orbit prediction accuracy is a function of the ability to determine the current state of the system as well as to characterize stochastic processes that may be driving the system. in this study, stochastic processes are included for the nongravitational accelerations, which have been characterized as exponentially correlated process noise. When dynamic. stochastic processes arc not included in the orbit determination, the orbit prediction problem is relatively straightforward. Measurements arc processed and an estimate of the current state is obtained. A deterministic mapping is performed to the epoch of interest and the orbit prediction error is determined principally by measurement errors. The situation is quite different when dynamic. stochastic processes are driving the system and this is generally the case for real systems. The stochastic nongravitational accelerations must be included in the mapping. As a result, short-term orbit prediction errors arc dominated by measurement errors and long-term predictions arc dominated by the stochastic accelerations. Therefore, the long-term spacecraft orbit prediction cannot be significantly improved by processing more accurate data or introducing new data types. Improvement must be obtained by designing the spacecraft to minimize these errors, including instrumentation on the spacecraft to measure nongravitational forces and developing more precise stochastic error models.

Table 6
SPACECRAFT ORBIT PREDICTION ERRORS

Prediction Epoch*	Radial (m)	Downtrack (m)	Crosstrack (m)
200-km orbit			
0 days	137	49	60
1 day	338	289	56
3 days	856	1,470	117
7 days	1,239	5,707	124
10 days	1,115	5,294	235
50-km orbit			
0 days	16	11	3.1
1 day	18	39	2.9
3 days	29	95	4.9
7 days	54	255	3.9
10 days	71	340	5.0
35-km orbit			
0 days	8.9	6.8	1.1
1 day	10	19	1.2
3 days	15	54	1.6
7 days	23	141	4.5
10 days	26	222	3.3

An orbit prediction time of at least a week is necessary to give the mission operations team time to plan science data gathering sequences and command the spacecraft to execute these sequences. The prediction time is measured from acquisition of the last data point, such as shattering the camera for optical data, to implementation of science sequences and includes time to process data, generate orbit determination solutions, and generate spacecraft command sequences, as well as round-trip light time. During this time, the spacecraft orbit error resulting from nongravitational accelerations must grow at a rate that is acceptable for science observations.

Science Instrument Pointing

For science instrument pointing, the location of a point on the surface of the asteroid relative to the spacecraft is of primary interest. Thus, the error in science instrument pointing attributable to navigation is a function of both the spacecraft orbit and Eros' attitude determination.

The estimated science instrument pointing errors are given in Table 7 for the same three orbit cases discussed above. These errors were obtained by combining the spacecraft orbit prediction errors given in Table 6 with the above attitude estimation errors. in general, the pointing errors at first increase as the spacecraft is maneuvered closer to the asteroid. This increase in pointing error is due to the inverse dependence of pointing on range for a fixed spacecraft position error. A maximum error of about one degree is observed for the 200 km orbit. As the spacecraft orbit is further lowered to 35 km, the predicted pointing errors decrease to fractions of a degree as the improved spacecraft orbit prediction capability overwhelms the effect of range. The science instrument pointing predictions appear to be well within tbc instrument fields-of-view throughout the orbit phase. The only exception may be for a few days following large orbit control maneuvers.

Table 7
SCIENCE INSTRUMENT POINTING ERRORS

Prediction Epoch*	Pointing error (deg) 200-km orbit	50-km orbit	35-kin orbit
0 days	1.7 X10-2	1. IX1O-2	1.1×10^{-2}
1 day	4. OX1O-2	4.6×10^{-2}	3.4 X1 O-2
3 days	1.8×10^{-1}	1.1 XI O-l	9.2 X10-2
7 days	1.2X 10^{+0}	3.1×10^{-1}	2.3×10^{-1}
10 days	9.7 X1 O-1	3.8x 10-]	3.6× lo-]

* Time from last data point used in solution

Eros Attitude Prediction

The above covariance analysis of spacecraft orbit prediction includes estimates of the initial errors in the attitude of the asteroid. The attitude error covariance may be mapped to various epochs that may be of interest. Science observations require estimates of the target-relative position vector and covariance. These may be obtained by simple transformation of the spacecraft state and asteroid attitude, The complete 12 by 12 covariance is required for this transformation.

Eros attitude estimation errors derived from a detailed covariance analysis are shown in 'l'able 8 for orbits with periapsis radii of 200 km, 50 km and 35 km. The attitude errors range from about 0.015 deg to 0.1 deg as given in terms of the pole and prime meridian. These values arc well within those required for navigation and science instrument pointing.

Table 8
EROS ATTITUDE PREDICTION ERRORS

Prediction Epoch*	Prediction error (1 sigma) α (deg)	δ (deg)	W (deg)
<u>50-km orbit</u>			
O days	1.5 X10-"2	I.3X1O-2	3.1 x 0^{-1}
1 day	1.2 X1O-"2	2.5x 10^{-2}	3.1 x 0^{-1}
3 days	6.8x10-"3	3.2x 10^{-2}	3.1 x 0^{-1}
7 days	2.6 X1 O-2	I.4X1O-2	3.2x 0^{-1}
10 days	1.4 X10-"2	1.5×10^{-2}	3.2x 0^{-1}
<u>35-km orbit</u>			
O days	9.1 X10-"3	6.5X 10^{-3}	8.6× 0^{-2}
1 day	6.5x 10-3	I.4X1O-2	8.7X 0^{-2}
3 days	3.1 XI O-3	1.8x 10^{-2}	9.0× ()-2
7 days	I.5X1O--2	6.7x 10-3	9.6X 0^{-2}
10 days	8.0x 10^{-3}	7.8x 10^{-3}	1.0× ()-1

* Time from last data point used in solution

Estimation of Physical Parameters of the Asteroid

The above covariance analysis of spacecraft orbit prediction errors also includes evaluations of errors in the physical properties of the asteroid. These errors are given in Table 9 for spacecraft orbit periapsis radii of 200 km, 50 km and 35 km. Also given in Table 9 arc the nominal values of these parameters.

The axes of the asteroid are the principal axes of inertia of Eros. The landmark cation errors given in Table 9 are for atypical landmark. This landmark has cartesian position coordinates $R_{\lambda x}$, $R_{\lambda y}$, and $R_{\lambda z}$. The landmark location errors indicate the accuracy with which surface features may be tied together on a map.

Table 9
EROS PARAMETER ESTIMATION ERRORS

Parameters	Nominal Values	Errors (1σ) Orbit Size 200 km	50 km	35 km
Landmark locations				
$R_{\lambda x}$ (m)	4-890	302	15	6.7
$R_{\lambda y}$ (m)	–71	9.3	23	6.3
$R_{\lambda z}$ (m)	--9,727	72	49	2.7
Mass properties				
m (kg)	1.330×10^{16}	1.2×10^{13}	3.8×10^{13}	1.8×10^{13}
GM (km^3/s^2)	8.865×10^{-4}	7.5×10^{-7}	2.5×10^{-6}	1.2×10^{-6}
I_{xx} (km^2)	22.8	CO	2.0×10^{-2}	6.3×10^{-3}
I_{yy} (km^2)	63.9	∞	2.0×10^{-2}	6.3×10^{-3}
I_{zz} (km^2)	70.9	∞	2.6×10^{-3}	7. OX IO-3
Gravity harmonics				
C_{20}	-3.03×10^{-2}	1.3×10^{-2}	9.2x 10-3	1,8 X1 O-3
C_{22}	$+3.78 \times 10^{-2}$	I.4X1O-2	6.5×10^{-4}	1.8x 10-4
C_{30}	$+1.05 \times 10^{-4}$	5.2x 10-2	2.2 X10-4	2.8x 10-5
C_{40}	$+4.09 \times 10^{-3}$	3.8x 10-]	1.6x 10-2	I.5X 10-3

The mass of the asteroid may be determined with great precision. This result is not surprising since this parameter has the strongest signature in the Doppler data among the various physical parameters and is consistent with results obtained on other missions. For the same reason, the low-degree gravity harmonics may be estimated with moderate precision; however, this determination deteriorates rapidly for larger orbit sizes or for higher degree harmonics.

The determination of the elements of the inertia tensor of the asteroid is critical to spacecraft orbit determination and prediction of the asteroid attitude. The moments of inertia about the principal axes are also of scientific interest since they provide some insight into the internal mass distribution. It is well known that the internal mass distribution and consequently the moments of inertia of the asteroid cannot be determined uniquely by external measurement of the gravity field. Also, the moments of inertia cannot be determined uniquely by observation of the tumbling of a free body in inertial space. Two objects of the same shape and uniform density will tumble in the same way independent of their size just as two bodies of different masses will accelerate at the same rate in a gravity field. However, the moments of inertia may be determined by combining observations of the asteroids rotation with the gravity field determined by observations of the spacecraft's motion.

The moment of inertia estimation error is also influenced by the amount of free precession experienced by the asteroid. The greater the amount of free precession, the smaller these estimation errors become. For the case of rotation about a principal axis, the spin axis is fixed in inertial space, and the angular accelerations are zero. This makes an estimate of moments of inertia indeterminate. For free precession down to about a tenth of a degree, the moments of inertia may be determined to less than one percent.

Gravity harmonics are in themselves of interest to science. When compared with the asteroid shape, some insight may be obtained into Eros internal structure. The location of the center of mass derived from the first degree harmonic coefficients give a direct indication of overall mass distribution. The second degree harmonic coefficients relate to the radial distribution of mass when combined with the rotational dynamics to estimate the moments of inertia. Higher degree harmonics may be compared with surface features to gain additional insight into mass distribution. For the limited data sets used to generate the results shown in Table 9, estimates of Eros's gravity field through degree four are obtained. It is expected that processing longer data arcs at low altitude, including polar orbits, will yield an accurate eighth degree gravity field.

SUMMARY AND CONCLUSIONS

This paper has presented a description of the navigation of the Near Earth Asteroid Rendezvous mission. During the post-rendezvous phases, accurate orbit determination is needed to support trajectory control maneuvers and science instrument pointing. Determination of the orbit of a spacecraft about an asteroid requires the development of an accurate model of the asteroid and the spacecraft flight environment.

The data types for orbit determination include Doppler tracking o?' the spacecraft, optical imaging of the asteroid and laser altimetry. A new development is the tracking of landmarks. This technique results in considerable improvement in orbit determination accuracy over simply tracking the center of figure. Another new data type is laser altimetry. 'l'his data type will provide useful information for developing a precise shape model and provide a backup for optical navigation in the event of a camera failure.

The effect of nongravitational accelerations resulting from outgassing or solar radiation pressure on spacecraft orbit prediction is described. It is shown that short-term predictions of a day or so are dominated by measurement errors whereas long-term predictions of several days are dominated by nongravitational accelerations. The orbit prediction errors grow from tens of meters to several hundred meters over ten days.

Spacecraft orbit determination errors are given for various orbit sizes ranging from 20 km to 200 km. The results are dominated by optical measurement errors when the spacecraft is relatively close to the asteroid and increase as the size of the orbit is increased. 'J'his increase may be attributed to weakening of the gravity harmonic signatures in the Doppler data. For tbc orbit sizes of primary interest, ranging from 35 km to about 100 km, the orbit determination errors vary from 20 to 50 meters for a l-day prediction and up to 400 meters for a 10- day prediction.

Results are also given for Eros attitude estimation and prediction. The attitude of the asteroid is of interest for science instrument pointing. These results tend to follow the same trends as the spacecraft orbit determination results, and attitude estimat ion errors vary from 0.015 deg. to 0.1 deg. over the range of orbit sizes from 35 km to 100 km.

Estimation of the spacecraft orbit and Eros attitude results in the incidental determination of many parameters that describe the asteroid and are of interest for science. Mass properties and the gravity field are determined, all of which are of considerable interest to the science team. 'l'he moment of inertia determination is of particular interest since it directly relates to the internal mass distribution of the asteroid. It is shown that the moments of inertia may be determined by observations of the asteroid's free precession, in conjunction with the determination of it's gravity harmonics.

ACKNOWLEDGEMENT

The work described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

REFERENCES

1. Farquhar, R., et. al., *Special Issue on the NEAR Mission to 433 Eros, The Journal of the Astronautical Sciences,* Vol.43, *No. 4, 1995* (in press).
2. Miller, L. J., Miller, J. K., Kirhofer, W.E., "Navigation of the Galileo Mission", Paper AIAA 83-0102, 1983.
3. Weeks, C. J., "Orbit Determination for the Mariner Mark II Cornet I{endczvous/Asteroid Flyby Mission: The Orbiting Phase," *Advances in the Astronautical Sciences: Astrodynamics 1985*, Vol. 58, l't. 11, Kaufman, B., et al., cd., Univelt, San Diego, 1986, pp. 1045-1064.
4. Miller, J. K. and Wood, L. J., "Navigation of the Mariner Tempel 2 Mission: Orbit Phase and Penetrator Deployment," AIAA Paper 87-0092, Al AA 25th Aerospace Sciences Meeting, January 1987, Reno, Nevada.
5. Miller, J. K., Weeks, C. J., Wood, L.J., "Orbit Determination of the Comet l{cndczvous/Asteroid Flyby Mission: Post-Rendezvous l'hascs", 27th Aerospace Sciences Meeting, Reno nevada, January 1989.
6. Miller, J. K., Weeks, C. J., Wood, L.J., "Orbit Determination Strategy and Accuracy for a Comet Rendezvous Mission", *Journal of Guidance, Control and Dynamics,* Vol 13, *No 5,* September-October 1990, pp 775-784.
7. Scheeres, 1). J., et. al.,''Navigation for Low-Cost Missions to Small Solar-System Bodies>', lAA-1,-03021', April, 1994.
8. Hamilton, T. W. and Melbourne, W. G., "Information Content of a Single Pass of Doppler Data from a Distant Spacecraft," *The Deep Space Network Space Programs Summary 3'7-39,* Vol. 111, Jet l'repulsion laboratory, Pasadena, California, 31 May 1966, pp. 18-23.
9. Gaskell, R.W., "Digital identification of Cartographic Control Points", }'holographic Engineering and Remote Sensing, Vol. 54, No. 6, Part 1, June 1988.
10. Owen, W.M. and Yeomans, D. K., "The Plate Overlap Technique Applied to CCI) Observations of 243 Ida". Astronomical Journal, 1994, (in press).
11. Bierman, G. J., *Factorization Methods for Discrete Sequential Estimation,* Academic Press, Inc., New York, 1977, pp. 171-178.
12. Miller, J. K., "A Continuous Square Root Information Filter- Smoother with Discrete Data Update," AAS/AIAA Spaceflight Mechanics Meeting, February, 1994.
13. C. L. Potts, M. G. Wilson, "Maneuver Design for the Galileo VEEGA Trajectory", A AS Paper 93-566, AAS/AIAA Astrodynamics Specialist Conference, Victoria, B. C., Canada, August 16-19, 1993.
14. Standish, E. M., J} 'I, Interoffice Memorandum, 314.10-109, January 20,1995.
15. Marsden, B. G., Sekanina, Z., and Yeomans, D.K., "Comets and Nongravitational Forces. V.", Astronomical Journal, 1973, Vol. 78, pp. 211-225.
16. Yeomans, D. K., JPL Interoffice Memorandum 314.10-65 to B.G. Williams, January 31,1993.
17. Yeomans, D.K., "A Review of Cornets and' Nongravitational Forces". Proceedings of conference on Asteroids, Comets, Meteors, 1994 (in press).
18. Scheeres, 1). J., "Short Term Satellite Dynamics About Comets: Outgassing and Solar Pressure Effects", JPL EM 314-572 (internal document), September 15, 1993.
19. Scheeres, D. J., "Satellite Dynamics About Tri-Axial Ellipsoids", in *Proceedings of Advances in Nonlinear Astrodynamics,* E. Belbruno Editor, Geometry Center Preprint No. GCG65, 1994.
20. Scheeres, D. J., "Satellite Dynamics About Asteroids", Paper A AS *94-112,* 1994.

21. Magnusson, P., Barucci, M. A., Drummond, J.D., Lumme, K., Ostro, S. J., Surdej, J., Taylor, R.C., and Zappala, V., "Determination of Pole orientations and Shapes of Asteroids", Asteroids II (edited by R.P. Binzel, T. Gehrels, and M .S. Matthews), University of Arizona Press, 1989, pp. 66-97.

22. Millis, R.L. and Dunham, D. W., "Precise Measurements of Asteroid Sizes and Shapes from Occultations", Asteroids II (edited by R.P. Binzel, T. Gehrels, and M .S. Matthews), University of Arizona Press, 1989, pp. 148-170.

23. Ostro, S.J. Campbell, D.B. Chandler, J. F. Shapiro, I.I., Hine, A. A., Velez, R., Jurgens, R.F., Rosema, K.D., Winkler, R., and Yeomans, D. K., "Asteroid Radar Astrometry", Astronomical Journal, Vol. 102, 1991, pp. 1490-1502.

24. Ostro, S. J., "Planetary Radar Astronomy", Review of Modern Physics, Vol. 65, No. 4, 1993, pp. 1235-1279.